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DESIGN STUDY OF A LARGE UNCONVENTIONAL LIQUID PROPELLANT ROCKET ENGINE AND VEHICLE

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Final Report
Report No. LRP 257

Volume 1: Summary

Contract NAS 5-1025

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FOREWORD

This is the final report of the work accomplished under Contract NAS 5-1025 (Design Study of a Large Unconventional Liquid Propellant Rocket Engine and Vehicle) and constitutes complete fulfillment of the Task 6 (reports) requirement.

To lend more effective meaning to the full scope of the investigation originally envisioned, much data as well as design work generated under company sponsorship and other programs has been applied and is referred to throughout the text. In this way, it is possible to provide far greater achievement and comprehensiveness in an effort of this nature within the limitations of its funding.

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I. CONCLUSION

The results of this investigation show the desirability and feasibility of future launch vehicles with single-stage-to-orbit capability. Incorporating the advanced engine, described later, makes the orbital booster even more attractive.

The vehicle shown in Figure I-1 was sized on the basis of two-million-lb sea-level thrust. However, the concept can be applied over a range of thrust levels of at least two- to 24-million lb. This vehicle incorporates an advanced engine which operates at 2,500 psi thrust-chamber pressure, uses a staged combustion cycle, and an altitude-compensating nozzle. Liquid-oxygen and liquid-hydrogen propellants are used. The following is a brief discussion supporting this conclusion.

One of the most important features of an earth-to-orbit vehicle is its number of stages. Thus, the orbital booster offers improvements in booster vehicle operational simplicity, reliability, and cost. Also, an orbital booster would make possible the recovery of the entire earth-to-orbit-booster system using a single recovery device. This gives the best recovery economy of all methods considered. Further, it precludes the possibility of stages falling in populated areas, thus making possible eastward launches from the west coast or the interior.

Orbital booster development cost is low and inherent reliability is high because there is only a single set of tanks, frames, and engines to develop. Also, compared to multistage boosters, the orbital booster has no in-flight engine starts, no interstage separation devices, no interstage retro rockets, less flight instrumentation, fewer malfunction detection systems, fewer ordnance systems, and less ground support and checkout equipment.

I, Conclusion (cont.)

As mentioned previously, the orbital booster becomes particularly attractive when propelled by an engine incorporating certain advanced features. The best engine evolved during this investigation is shown in Figure III C-1 and is described as follows.

The advanced engine has eight thrust chambers, circumferentially located around the turbopump which is situated in the center of the engine. These chambers feed into a common forced-deflection nozzle through eight transition nozzles giving a smooth transition for the supersonic gases. Thrust takeout to the airframe is provided through a cone (not shown) which attaches to the engine at the main nozzle near the circle where the transition nozzles and main nozzle join. Thrust-vector control is provided by high-pressure gases bled from the first combustor and injected radially inward through the nozzles shown.

I, Conclusion (cont.)

A. MAJOR COMPONENTS, ADVANCED ENGINE

1. Cycle

Staged combustion, fuel-rich gas generators, parallel hydrogen flow-through gas generators, and thrust-chamber coolant jacket.

2. Thrust Chambers

Eight combustors; mixture ratio of 6:1; chamber pressure of 2,200 psia; hydrogen-cooled chamber walls; altitude compensating, eight throats, single skirt, forced-deflection nozzle; vacuum area ratio of 125:1.

3. Hydrogen Turbopump

A single inlet: two-stage, centrifugal-flow pump directly driven by a two-stage, parallel-flow gas turbine that also drives the oxygen pump; pump discharge pressure of 4,100 psia.

4. Oxygen Turbopump

A main-stage turbopump consisting of a single inlet, single-stage, centrifugal flow pump; low shaft speed inducer pump stage precedes the main-stage turbopump; pump discharge pressure of 4,100 psia.

I, A, Major Components, Advanced Engine (cont.)

5. <u>Gas Generators</u>

Eight, fuel-rich, side-by-side generators; mixture ratio of 1.1:1; chamber pressure of 3,900 psia; flash-over ports between generators for ignition redundancy.

6. <u>Thrust-Vector Control</u>

Secondary gas injection using gas generator products.

I, Conclusion (cont.)

B. CONCLUSIONS AFFECTING ENGINE AND VEHICLE DESIGN

The following conclusions, affecting engine and vehicle design, are based upon overall study results.

1. The cost and reliability of a rocket vehicle are heavily influenced by its number of stages. It becomes more feasible to eliminate stages if inherently high specific impulse and low vehicle inert parts weight can be achieved. Thus, concepts offering advantages in either of these areas provide the greatest promise for reducing costs and increasing reliability. This conclusion was substantiated by many of the findings that follow and which were arrived at independently.

2. Use of multiple rather than single components is not generally recommended for achieving low production cost, light weight and high reliability. The combustion chamber and gas generators are exceptions because significant development cost savings are available by segmenting these components.

3. Rigidly mounting the engine to the airframe offers potentially lower structural weights.

4. Thrust-vector control by secondary gas injection is best.

5. Recovery of the first stage can result in significant savings only if this recovery is effected from a high burnout velocity. Recovery of a single-stage vehicle from orbit is best. Also desirable is recovery from booster burnout velocities between 12,000 and 20,000 ft/sec. Recovery of boosters from 7,000 ft/sec burnout velocity shows no advantage. This results because the necessary second-stage, which is not recovered, is quite large and expensive.

I, B, Conclusions Affecting Engine and Vehicle Design (cont.)

6. Two-stage vehicles are superior to three-stage vehicles for the escape mission. For the orbital mission, the single-stage oxygen/hydrogen vehicle is best.

7. Oxygen/hydrogen propellants are best.

8. Rocket engine and stage hardware production costs/lb of thrust decrease as thrust is increased within the range of this investigation.

9. Pump-fed rockets are superior to pressure-fed rockets for booster stages within the thrust range of this investigation.

10. The first stage of a two-stage launch vehicle should provide the greater portion of the mission ideal velocity.

11. No significant advantage is obtained from unconventional types of airframe construction if aluminum is used.

12. Materials such as titanium and filament plastics or metals offer some promise for significantly reducing airframe structural weights; however, investigations are needed to determine how to best utilize the high strength properties of these materials.

(End I)

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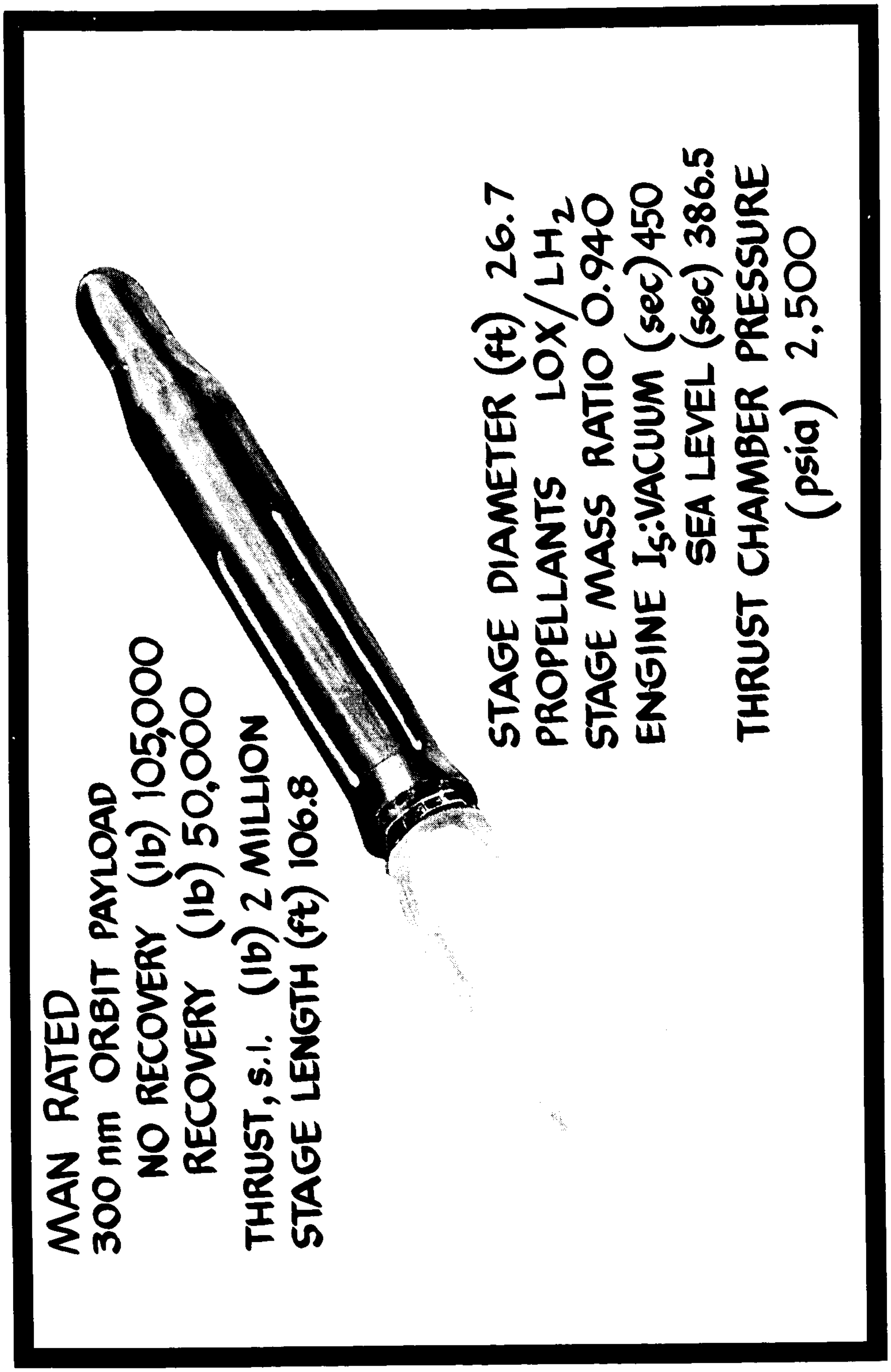

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Orbital Booster

Figure I-1

[redacted]

II. RECOMMENDATIONS

It is recommended that programs be initiated immediately to show feasibility and obtain design data for the advanced engine described. The programs required to accomplish this are delineated in Volume III, Section III and are summarized in Section III of this volume.

An additional recommendation is in the area of recovery. The desirability of the recoverable orbital booster was shown during this investigation. However, because of a lack of information, little could be concluded as to the feasibility of such a scheme. Therefore, it is recommended that a detailed investigation be undertaken to determine the feasibility of orbital booster recovery. Then on the basis of the results acheived, engine and vehicle requirements can be reviewed and orbital booster recovery re-evaluated.

(End II)

III. SUMMARY

A. GENERAL

The overall purpose of this program was to determine if an unconventional liquid-chemical rocket engine and vehicle design concept could be developed and produced for a large launch stage of two-million-lb thrust, or more. This was to be accomplished at a significantly reduced cost while assuring increased reliability when compared with current conventional engine vehicle designs. While the primary area of interest was determining an engine concept, it was necessary that combined engine-vehicle configurations be explored to fully evaluate the advantages offered by the various engine concepts.

A design was developed that is superior to the conventional engine. Establishment of the design criteria as well as the engine geometry was then undertaken (secondary program objective).

Liquid oxygen/liquid hydrogen and liquid oxygen/hydrocarbons propellant combinations were considered.

Program effort was divided into the following six tasks:

1. Task I

Technical evaluation of various unique concepts and selection of the best engine-vehicle system.

2. Task 2

Evaluation of the best system to determine the optimum values for major parameters.

III, A, General (cont.)

3. Task 3

Delineation of feasibility programs required to move the best advanced system towards reality.

4. Task 4

Establishment of cost scaling factors ranging to 24-million-lb thrust for the advanced system.

5. Task 5

Upper-stage application of the advanced system.

6. Task 6

Reports requirement.

Task 1 was accomplished during the first four months of the program. The remaining tasks were completed in the following six months. Individual task efforts are summarized in the following discussions.

(End III A)

III, Summary (cont.)

B. TASK 1

1. Concepts Considered

A large number of vehicle and engine concepts were generated at the outset of this investigation. These included original concepts and variations of previously considered concepts.

The scope of this investigation is best shown by the following list, which includes the most important concepts considered.

a. Staging: Tandem, parallel, partial.

b. Clustering: Complete propulsion systems, subsystem, components.

c. Propellant feed: Pressure, pump.

d. Engine cycles: Gas generator, staged combustion, heated hydrogen bleed, heated hydrogen topping, third fluid, delivery feed, gas generator with turbine exhaust afterburner.

e. Propellants: O_2/H_2, O_2/RP-1.

f. Engine configurations: Conventional, single turbopump, clustered thrust chamber, plug, forced deflection.

III, B, Task 1 (cont.)

g. Thrust-vector control methods: Gimbaled engine, jet vanes, fluid injection, throttled combustion chamber in multichambered configurations.

h. Pressurization system: Stored gas (heated and nonheated), heated propellants, gas generator (solid, liquid), VāPak.

i. Combustion chamber: Single as opposed to multiple, round as against annular.

j. Nozzles: De Laval, plug, forced deflection.

k. Turbopumps: Inducers, single as compared with clustered, integrated turbopump system, gear-driven as opposed to direct, axial flow in relationship to centrifugal flow pumps, positive displacement pumps, jet pumps.

l. Valves: Butterfly, gate, spring, plug, ball, ring gate, pump-mounted gate.

m. Actuation systems: Electrical, mechanical, hydraulic, pneumatic.

n. Turbine speed governors.

o. Airframe and tankage: Conventional as opposed to unconventional configuration, materials, cluster, rigid as against gimbaled engine, contoured propellant liner.

III, B, Task 1 (cont.)

2. Bases of Evaluation

The bases established for evaluating vehicle and component concepts are explained in the following discussion.

a. Methods for Vehicle Concept Evaluation

Concurrently with the generation of concepts, vehicle system cost and reliability investigations were initiated to determine where the greatest potential for improvement in rocket vehicle system was to be found. A cost model was evolved to accomplish this. Examination of this cost model revealed that there is a great deal of fixed cost associated with each stage of the rocket vehicle. Also, a substantial contribution to this cost/stage is its development. Therefore, it was concluded that concepts permitting elimination of stages, or use of currently developed stages would result in lowest cost vehicles. Obviously, the elimination of stages is also beneficial from a reliability standpoint.

(1) Three calculable factors were then evolved for rating the various vehicles that would provide a measure of the trends previously mentioned. These are:

(a) Cost/lb of payload
(b) Unreliability
(c) Payload/lb of second-stage thrust level.

Cost/lb of payload was calculated on the basis of the cost model described in Volume 2 for the orbit and escape mission. Other assumptions in these cost comparisons are detailed in the following paragraph. Unreliability was estimated by the number of stages (a single-stage vehicle has an

III, B, Task 1 (cont.)

unreliability of 1 and a two-stage vehicle has an unreliability of 2). Payload/lb of second-stage thrust was used to reflect the largest vehicle that could use an upper stage currently being developed, such as Saturn S-IV, S-IV B, or S-II. A vehicle with a high payload/lb of second-stage thrust can be designed for a higher payload without incurring second-stage development cost.

(2) It is recognized that relative values of cost/lb of payload are influenced heavily by the assumption used. Therefore, the assumptions which are the basis of the comparisons in Task 1 are as follows:

(a) Cost model, see Section III, D, Volume 2

(b) Missions, orbit, escape

(c) Number of flights, 25; 100; and 400

(d) Payload for all vehicles is the same. Base payloads were established for conventional two-stage oxygen/hydrogen vehicle using a two-million-lb thrust first-stage engine.

(e) Vehicle staging for minimum cost/lb was selected.

(f) Development cost for stages other than the first stage was not included.

(g) Production learning curve has an 85% slope.

(h) Propellant performance is based on shifting equilibrium.

(3) In considering booster recovery, the following assumptions were applied:

(a) Flights/booster, 10

(b) Recovery system cost, 0

III, B, Task 1 (cont.)

(c) Recovery system weight

1 Low velocity (7,000 ft/sec) recovery system weight = 0

2 Intermediate velocity (15,000 ft/sec) recovery system weight = stage inert weights

3 Recovery from orbit recovery system weight = 0.61 stage jettison weight. Payload to decrease by 6%.

The most controversial of these assumptions is the use of constant payload rather than constant thrust as the basis for determining cost/lb of payload. The most important difference in these two bases results because, for a given vehicle, cost does not increase proportionally with vehicle size, while payload does. As a result, the use of constant thrust criterion favors vehicles with high payload of takeoff weight ratios.

Task 1 analyses were conducted on the basis of the fixed-payload criterion. The fixed-thrust criterion was subsequently applied to the same vehicle types and the results compared (Appendix B). Results of the investigation were affected to some extent, but the basic conclusion remains valid that a single-stage-to-orbit vehicle using oxygen/hydrogen propellants and the advanced engine, is best.

To establish a base line, against which concepts could be evaluated, it was necessary to select a standard vehicle early in the program. The factors shown to be most influential in reducing costs and increasing reliability were incorporated insofar as current technology permitted. Upon this basis, a two-stage pump-fed oxygen/hydrogen vehicle was selected as the standard.

III, B, Task 1 (cont.)

Upon more careful analysis, it was found advisable to increase the first-stage ideal velocity increment. The resultant vehicle can use a Saturn S-IV B stage as its second stage. In addition, the booster has improved single-stage-to-orbit payload capability and can be adapted to either orbit or escape missions. Final system selection is based upon comparison with the best conventional vehicle.

Ultimate manned rating of all vehicles was assumed. The Boeing Company and the Martin-Marietta Corp. were consulted as to what vehicle requirements are imposed by a manned rating. (See Volumes 5 and 6 for results of these efforts.) Safety factors of 1.5 were used. In addition, neutral or near-neutral aerodynamic stability will be required to assure safe escape in the event of a vehicle control malfunction while in the atmosphere. Fully loaded, unpressurized, free-standing capability is necessary.

By placing the oxygen tank ahead of the hydrogen tank, and providing a slight flare in the engine skirt, aerodynamic neutral stability was attained with very little weight penalty. (See The Boeing Company study report, Volume 5.)

b. Method for Component Concept Evaluation

As indicated, the most significant cost effects were found by eliminating stages or by applying stages already developed. Elimination of stages also benefits vehicle reliability. Development of high performance in the booster makes both of these goals more easily attainable. High performance results from both high specific impulse and low inert weight. In general, the

III, B, Task 1 (cont.)

concepts considered involve both weight and specific impulse changes. To evaluate the combined effects of these changes, vehicle payloads were compared. The effect of specific impulse and inert weight were thus combined so that direct comparisons, on the basis of payload alone, were used to show relative component concept ratings.

3. Concept Evaluation

a. Vehicle Concepts

There are six basic vehicle comparisons that were performed. These were for the purpose of selecting propellants and for evaluating recovery methods, staging methods, clustering, pressure-fed booster, and the best vehicle.

Each of these evaluations are briefly discussed as follows.

(1) Propellant Selection

One-, two-, and three-stage vehicles using oxygen/hydrogen or O_2/RP-1 propellants, or combinations of both in different stages, were examined. Three-stage vehicles were shown to be inferior to the two-stage vehicles, even for the escape mission. Therefore, they were eliminated from further consideration.

Use of O_2/RP-1 in the booster was examined. This vehicle requires more thrust in the first and second stages than the two-stage oxygen/hydrogen vehicle for the same payload. In addition, O_2/RP-1 boosters

III, B, Task 1 (cont.)

do not have significant single-stage-to-orbit payload capability. Payload delivery cost is also somewhat higher for vehicles using O_2/RP-1 boosters. For these reasons, these vehicles are considered inferior to the two-stage oxygen/hydrogen vehicle. These comparisons are shown in Figures III B-1 and III B-2.

(2) Recovery Methods Evaluation

The following three methods of booster recovery were examined:

(a) Low velocity (ballistic)

(b) Intermediate velocity (winged vehicle)

(c) Recovery from orbit.

Applicable velocity ranges for each method and recovery system weight assumptions are given in Section III, B, 2. Results of the investigation are shown in Figures III, B-3 and III B-4. It was concluded from these figures that the best recovery method is the single-stage-to-orbit vehicle. This results because the entire vehicle is recovered.

If the cost comparison is made upon the basis of fixed first-stage thrust rather than fixed payload, none of the recovery methods shows significant cost advantage over nonrecoverable vehicles (Appendix B).

III, B, Task 1 (cont.)

(3) Staging Methods Evaluation

Staging methods were reviewed to determine if any offered significant advantage over tandem staging. Parallel staging, staging tanks only, and staging of engines only were examined. Results of this investigation showed that no significant reduction in cost or increase in reliability could be expected by selecting other than tandem staging.

(4) Evaluation of Clustering

An investigation was performed to determine if significant improvements in reliability and/or cost were to be gained through the application of clustering. This evaluation was not intended for the application to clustering of small control components. Such an evaluation is normally made during detailed design and is, therefore, beyond the scope of this investigation. The following applications of clustering were investigated.

(a) The development of one basic "building block" that can be applied to solving a multitude of payload and mission requirements.

(b) Avoiding problems such as tooling limitations that could occur if components become very large.

(c) Achievement of low cost through large quantity production of many small units.

(d) Obtaining reliability through the use of major component redundancy.

III, B, Task 1 (cont.)

The methods for clustering are numerous and include grouping of various numbers of engines, propellant tanks, turbopumps, combustors, and different combinations of these components. Because the potential number of clustering combinations is high, generalized approaches that would indicate trends were used in this analysis.

It was concluded as a result of this investigation that clustering, as discussed herein, offers no significant potential savings in cost or increases in reliability.

As indicated in Section III, A, 3, b, (1), significant benefit in lower development costs was found to result from clustering combustion components.

(5) Pressure-Fed Booster

An extensive examination of the pressure-fed booster concept was conducted during the six-million-lb thrust-level engine investigation prior to contract initiation. This study was considered conclusive to the point that investigation at the lower thrust level was not necessary. Several pressure-fed concepts are described in Section II, D, Volume 2. Design effort is summarized in Appendix H. An economic comparison is also shown in Section II, D, of Volume 2. Payload delivery cost comparisons clearly showed that the pump-fed oxygen/hydrogen vehicle is superior to the vehicle with the pressure-fed booster. Also, the pressure-fed booster has no apparent reliability advantage over a pump-fed stage. Furthermore, a pressure-fed booster alone cannot deliver payload to orbit. Therefore, this concept was eliminated from further consideration.

III, B, Task 1 (cont.)

(6) Evaluation of the Best Vehicle

The best vehicle concepts and the best engine component concepts were combined to provide the best unconventional vehicle, the orbital booster. This vehicle was described in Section I of this volume. Evaluations of this vehicle are shown in Figures III B-6 and III B-7. Note that significant advantages in cost have been elaborated in Section I of this volume.

b. Engine and Component Concept Evaluation

In view of the nature of the most beneficial factor for reducing cost and increasing reliability, as well as the comparisons previously shown between oxygen/hydrogen and O_2/RP-1 boosters, the investigative scope of engine and component concepts was modified. Efforts with high-thrust chamber pressure, which permits extraction of more useful energy from the propellants, was increased. Many engine cycles that more efficiently produce higher thrust chamber pressure were added. Effort expended in connection with other areas, such as oxygen/hydrocarbon propellants, was reduced.

As shown by the vehicle analyses, the single-stage-to-orbit oxygen/hydrogen vehicle appears attractive even if a conventional engine is used. Significant performance improvements would result in the single-stage-to-orbit vehicle being superior to other selections. Therefore, engine and component concepts were evaluated on the basis of single-stage-to-orbit payload capability. All unique concepts were then evaluated and those not contributing to higher performance were eliminated.

III, B, Task 1 (cont.)

(1) Four classes of concepts remained that showed promise for achieving study objectives. These are:

(a) Augmenting propellants with air

(b) Extracting more of the available energy from propellants

(c) Concepts resulting in inherently lighter stage inert parts weight

(d) Concepts showing clear advantages for lower development, production, and operational cost without sacrifice in performance.

In recent years, much effort has been expended on the first class of concepts. The most promising appears to be the Liquid Air Cycle Engine (LACE), whose feasibility is being intensively investigated under other contracts. Therefore, detailed analyses of air-breathers were not considered to be within the scope of this study. As a result, major emphases was placed upon the other classes of concepts--those producing higher specific impulse by extraction of more useful energy from the propellants, those showing promise for reducing stage inert weights, and those showing potential for reducing cost without sacrificing performance.

(2) Promising concepts within each of these remaining three classes are discussed as follows:

III, B, Task 1 (cont.)

(a) Higher Specific Impulse

Altitude compensating nozzle concepts, higher thrust chamber pressure, and several engine cycle concepts offer potential increases in available specific impulse.

Altitude compensation permits use of higher nozzle area ratio, yet it precludes exhaust gas separation in the nozzle, which could cause unpredicatable thrust-vector changes. Higher chamber pressure allows use of higher nozzle area ratio for a given nozzle size and weight. Engine cycles that have the potential for extracting more useful energy from turbine drive fluid result in higher efficiency and therefore, higher performance engines.

Results of an anlysis showing potential gains from the use of altitude compensating nozzles, higher than conventional chamber pressure, and various engine cycles are shown in Figure III B-8. Substantial increase in a single-stage orbital booster payload results when the best of these concepts are combined. In preparing Figure III B-8, the effect of increased chamber pressure upon turbopump weight was included. Also, turbopump work was subtracted from combustion gas enthalpy when calculating specific impulse for the staged combustion cycle.

The advantages in higher than conventional chamber pressure and the altitude compensating nozzle are evident in Figure III B-8. Two engine cycles were found to offer substantial vehicle performance increases over the gas generator cycle. These are the intermittent-delivery feed cycle and the staged combustion cycle. Both eliminate the mixture ratio degradation and substantially reduce the specific impulse degradation resulting from turbopump

III, B, Task 1 (cont.)

drive requirements at high pressure. Descriptions as well as technical analyses of these cycles are presented in Section II, E of Volume 2. While the intermittent-delivery feed cycle has the greatest potential for performance increase, no conclustion has been reached as to its technical feasibility. Therefore, the staged combustion cycle was selected as the most promising one.

(b) Reduced Stage Inert Parts Weight

The following were examined in the stage system to determine if lighter weight resulted when they were incorporated into a stage system.

1 Nozzle types

2 Engine configurations

3 Pump inducer

4 Large element injection

5 Rigid engine-airframe integration

6 Airframe configurations

7 Contoured suction lines.

Three basic nozzle-types were evaluated, De Laval, forced deflection, and plug. Methods for altitude-compensating De Laval nozzles proved to be cumbersome; therefore, emphasis was directed mainly towards the other two.

III, B, Task 1 (cont.)

Theoretically, the forced-deflection and plug nozzles give approximately equal performance. Therefore, engine configuration was examined to determine which nozzle type would provide the lightest weight for the engine. Engines using each of these nozzle types are shown in Figure III B-9. Detailed designs for both of these configurations showed that the forced-deflection design offered significantly lighter weight because hot-gas high-pressure components are more compact.

A pump inducer permits high turbopump rotating speeds and low tank pressures, both contributing to low weight. The use of a large element injector permits higher injection density, thus permitting use of smaller and lighter weight injectors. This weight savings accruing from the use of pump inducers and large element injectors is particularly significant in high chamber pressure engines.

As a result of design effort, it was found that airframe weight savings of approximately 6, 000 lb results if the engine can be rigidly mounted to the airframe. To fully evaluate this scheme, means for thrust-vector control, other than gimbaling the engine, were examined. The best method for thrust-vector control was found to be secondary gas injection. By incorporating secondary gas injection thrust-vector control and rigidly attaching the engine to the airframe, a net gain was obtained of nearly 4, 000-lb payload for the single-stage vehicle.

Novel airframe approaches in both configuration and materials were considered. The conventional tandem cylindrical tank configuration appears to be most suitable for the engines considered. There appeared to be no configuration offering any advantages over the conventional-type even when engine interactions were disregarded.

III, B, Task 1 (cont.)

Means for achieving weight reductions through use of materials other than aluminum for structure and tankage were then investigated.

In all of the designs shown, aluminum with a strength-to-density ratio of 645,000 in. was used. Titanium with a strength-to-density ratio of 760,000 in. has frequently been considered for propellant tanks. Use of this material would result in a 3% increase in the single-stage-to-orbit vehicle payload capability. Even higher strength-to-density ratio materials are theoretically feasible. Among these are the fiberglass and steel filaments. Theoretically, 3,000,000-in. strength-to-density ratio is attainable with these materials. A maximum weight savings of approximately 30,000 lb is available if this strength-to-density ratio is applied to the single-stage vehicle airframe inert weights. This would result in a 50% increase in single-stage vehicle payload capability. For recoverable vehicles, payload increases would be even more pronounced because in making the structure lighter, the recovery system also becomes lighter.

The vehicle requirement for neutral aerodynamic stability was imposed as one of the man-rating criteria. To accomplish this, the oxygen tank was located ahead of the fuel tank and a flared engine skirt was added. As a result of locating the oxygen tank forward, the oxygen propellant line contains approximately 13,000 lb of propellant. Most of this propellant can be used if cavitation at the tank outlet can be avoided. To avoid cavitation, funneling of the propellant line at the tank was suggested. This funnel section would be designed so that the propellant would increase in velocity at a lesser rate than the increase in the head above the propellant.

III, B, Task 1 (cont.)

(c) Low Cost

Use of single rather than multiple components was generally recommended as a result of the clustering study (Paragraph III, B, 3, a, (4)). In the case of combustors and turbopumps, this was further reviewed during the component investigation. Results of these studies are summarized as follows:

Combustor technology is developed largely through experimentation because there is no complete theory for combustion. Injectors up to 1.5-million-lb thrust have been demonstrated, but extensive development is required to achieve high performance. At higher thrust levels, development will be more expensive because of the greater fabrication and testing cost. Development can be minimized through the use of multiple rather than single combustors. On the other hand, turbopump technology is such that there is an excellent theory for the design of much larger turbopumps than existing rocket turbopump sizes. Investigations show that, for large engines, single turbopumps are superior to clustered turbopumps.

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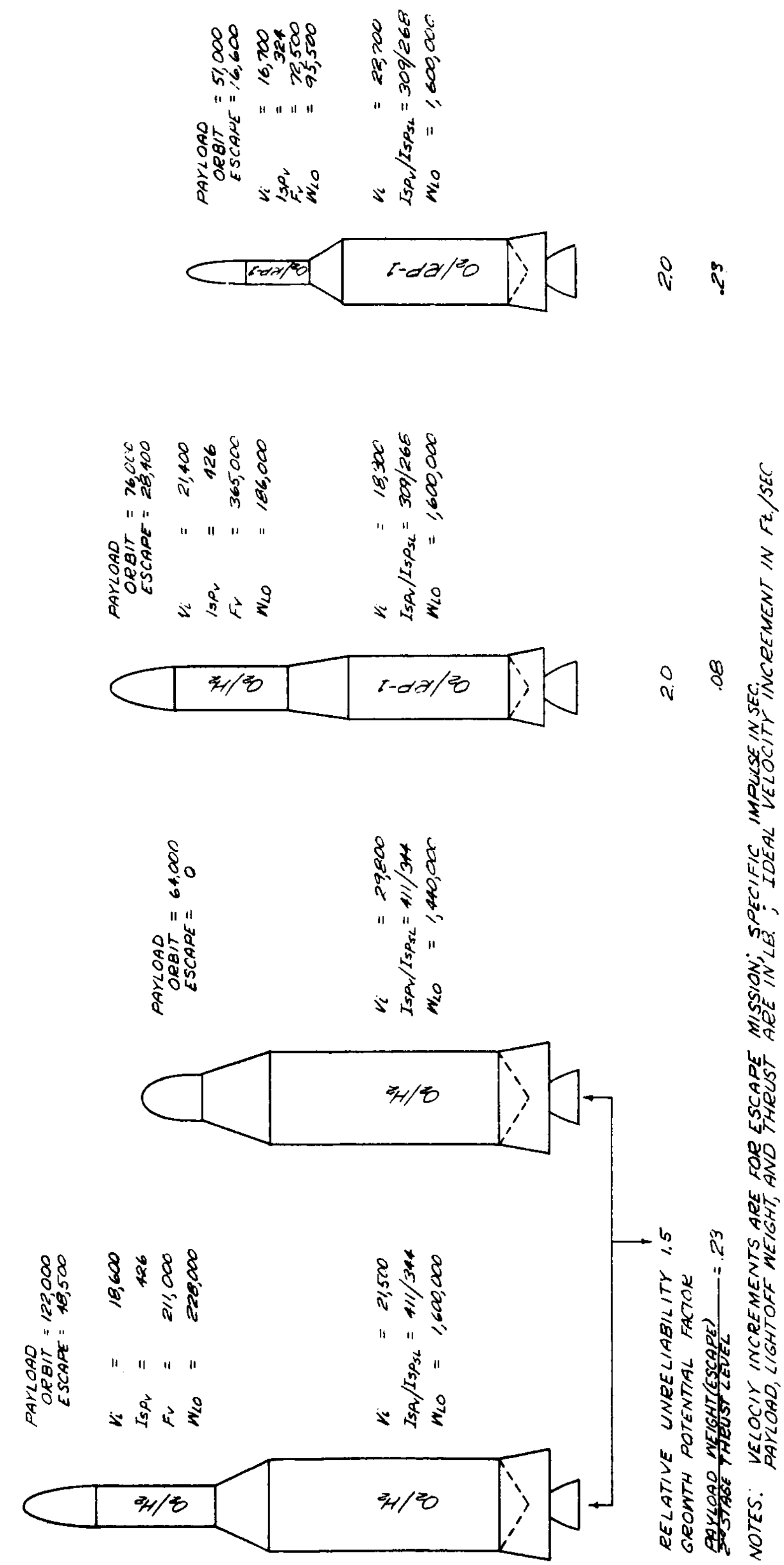

Propellant Comparison Configuration

Figure III B-1

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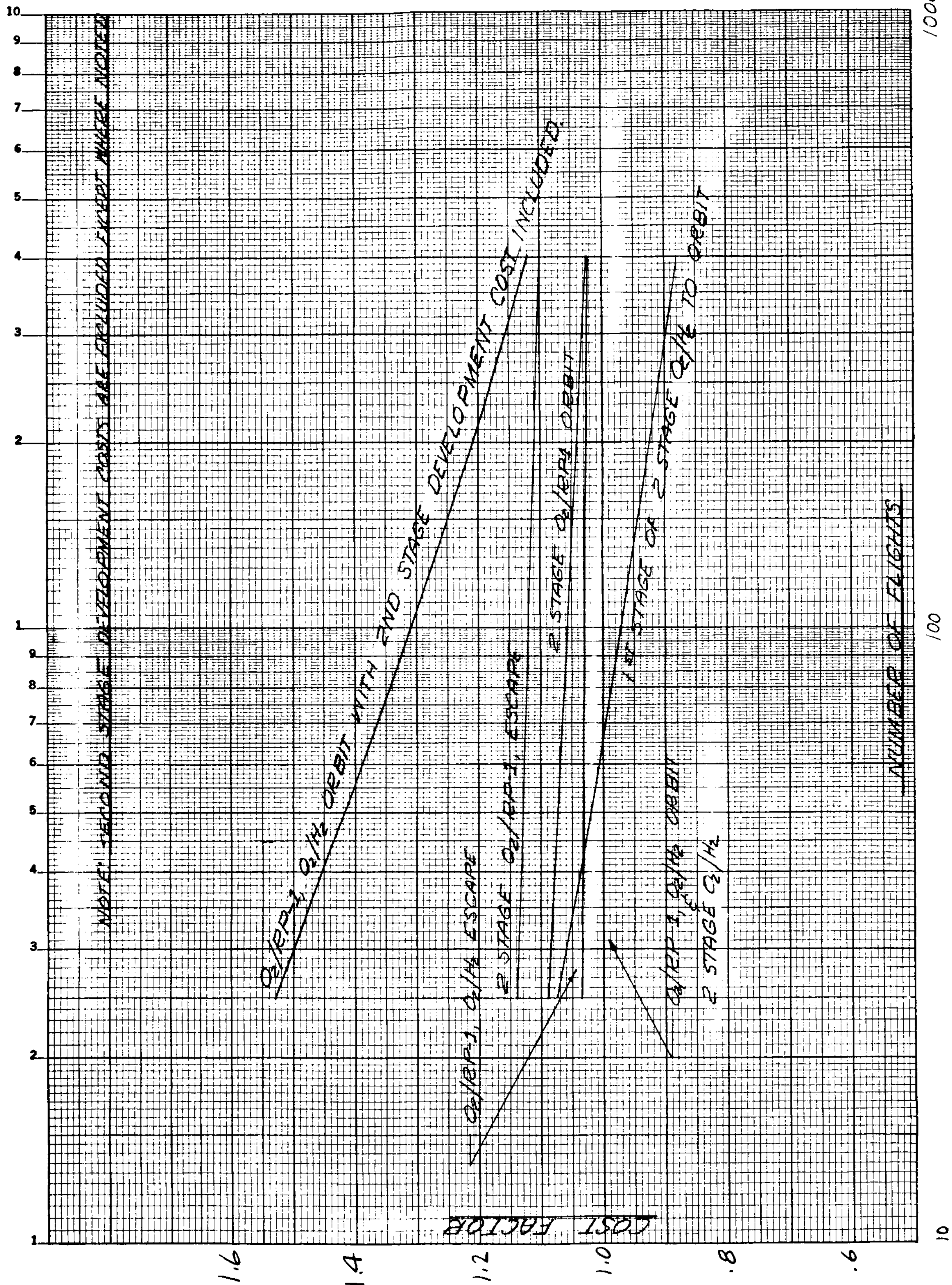

Propellant Comparison Cost Factor

Figure III B-2

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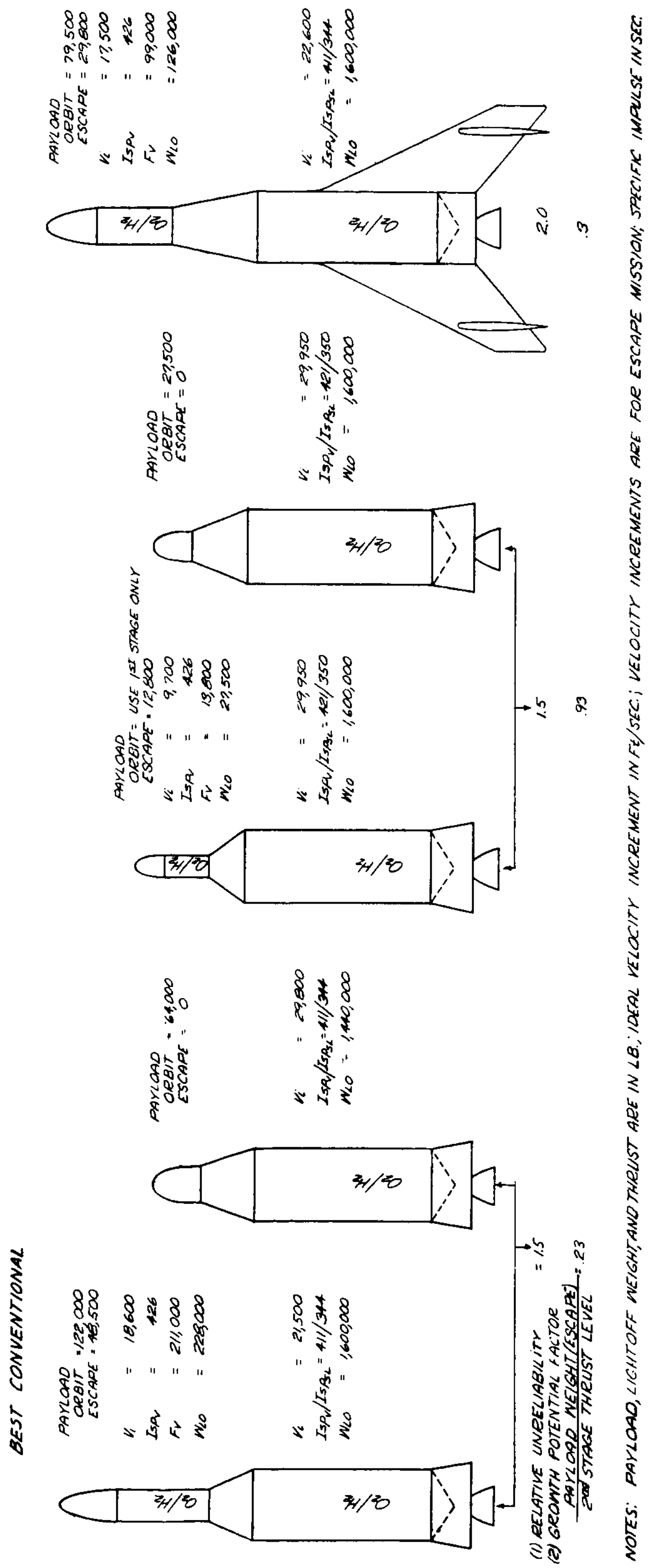

Recovery Methods Comparisons Configurations

Figure III B-3

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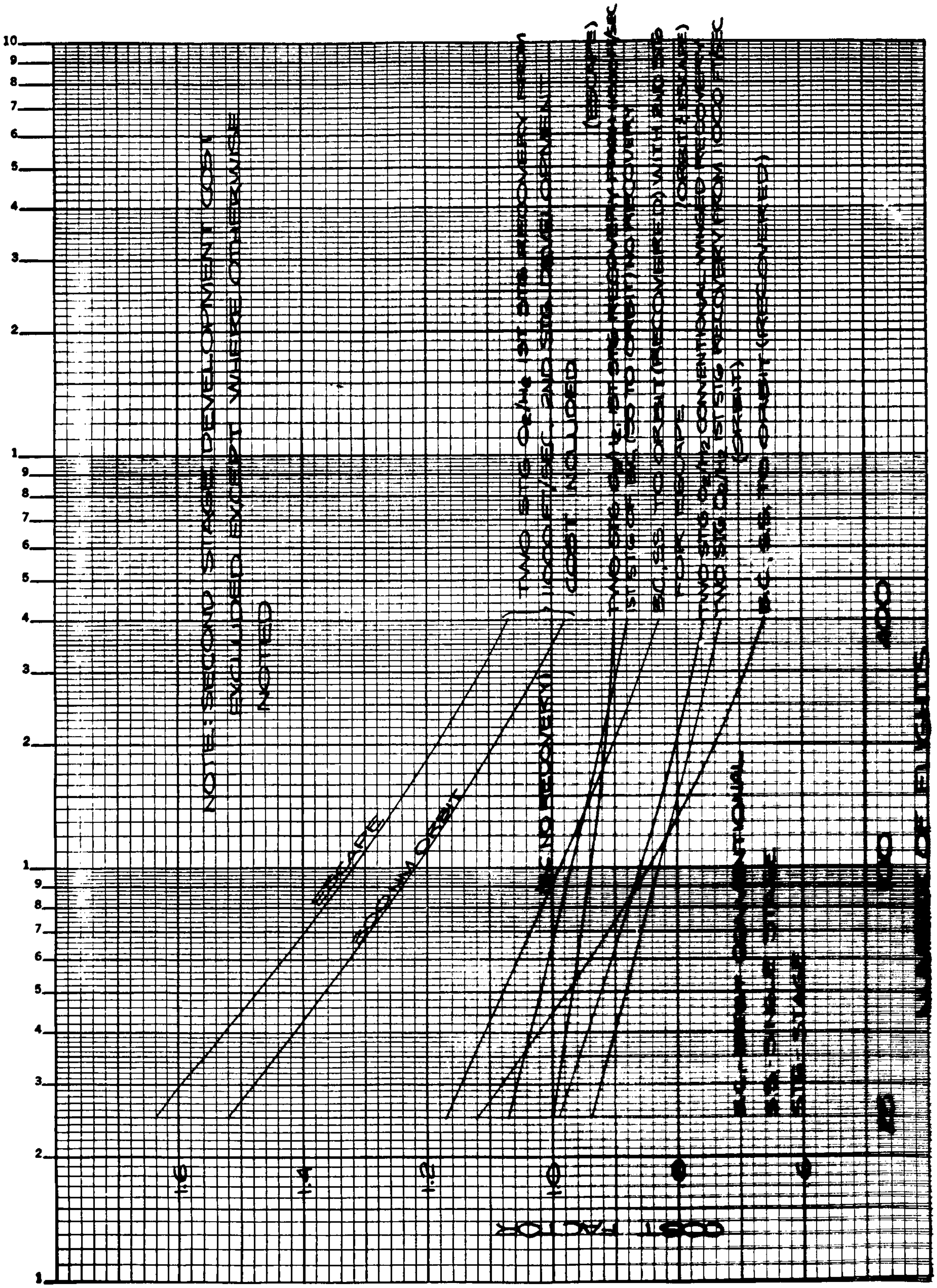
[redacted]



Recovery Methods Cost Comparisons

Figure III B-4

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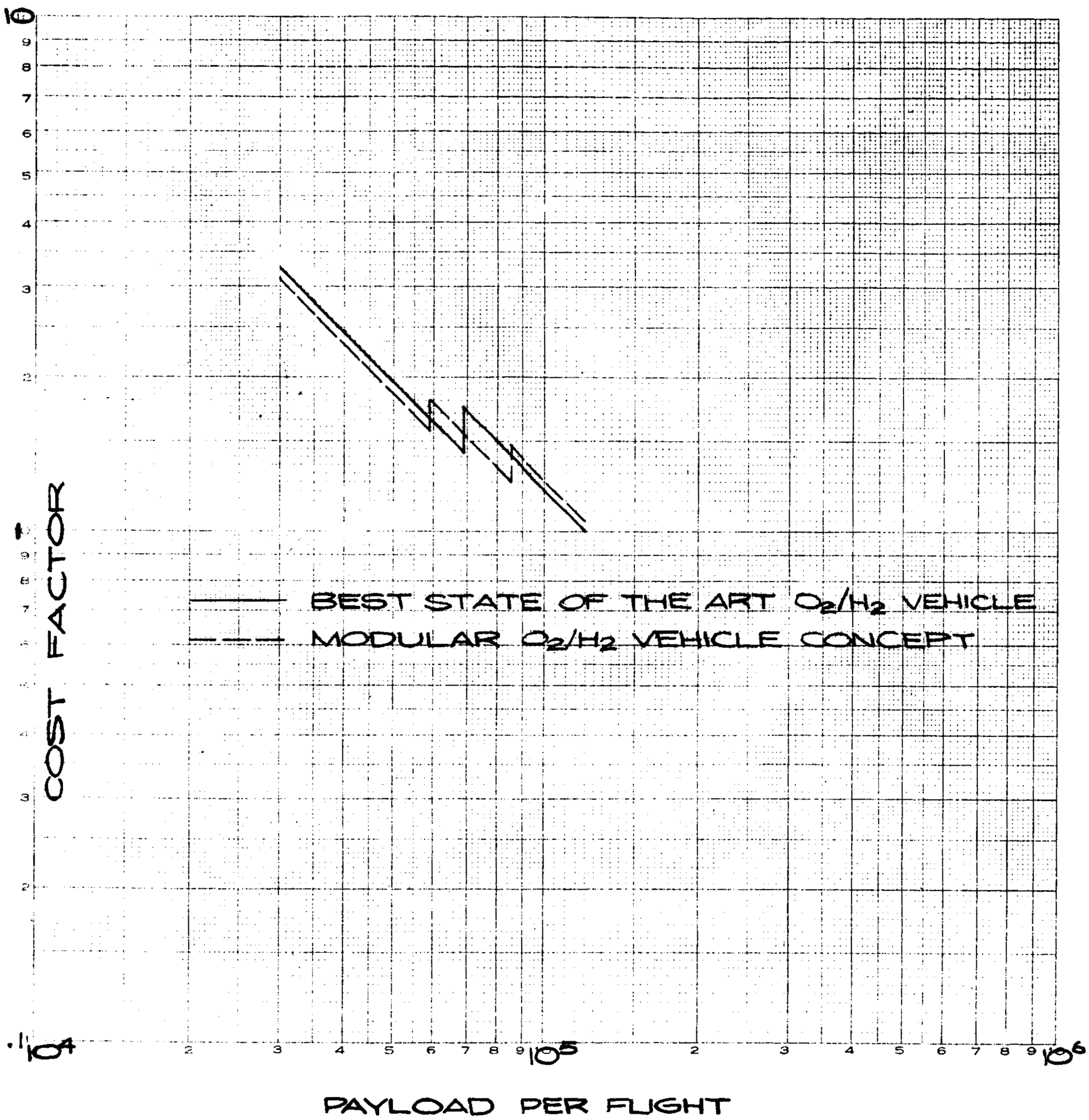
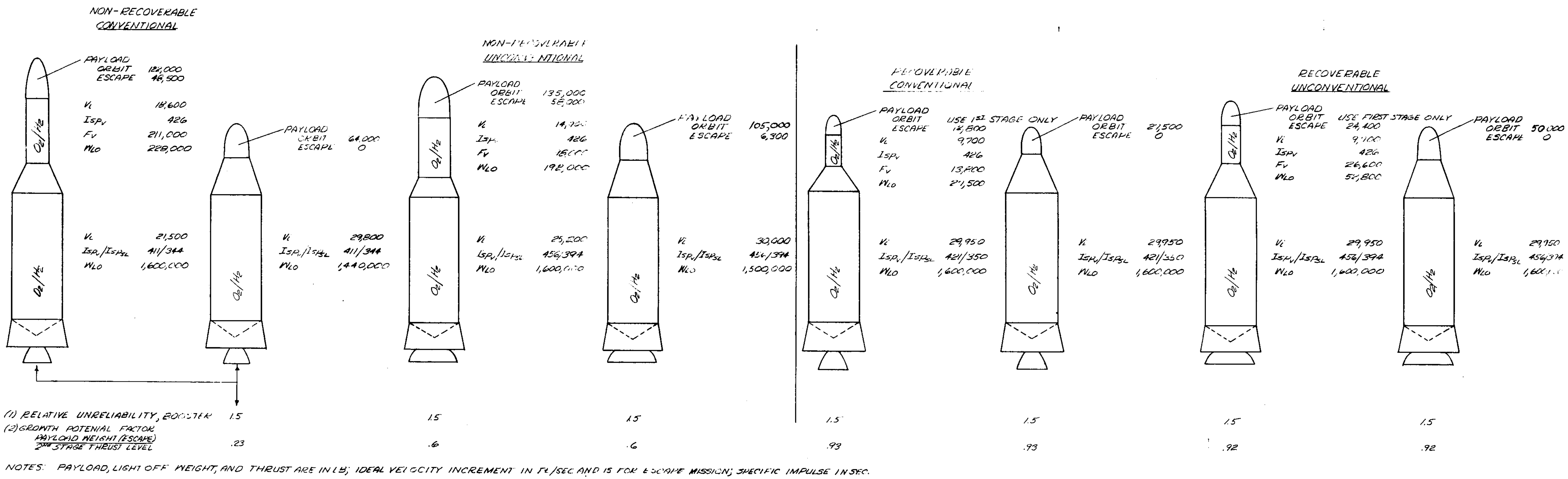


Cost Factor vs Payload per Flight

Figure III B-5



Unique vs Best Conventional

Figure III B-6

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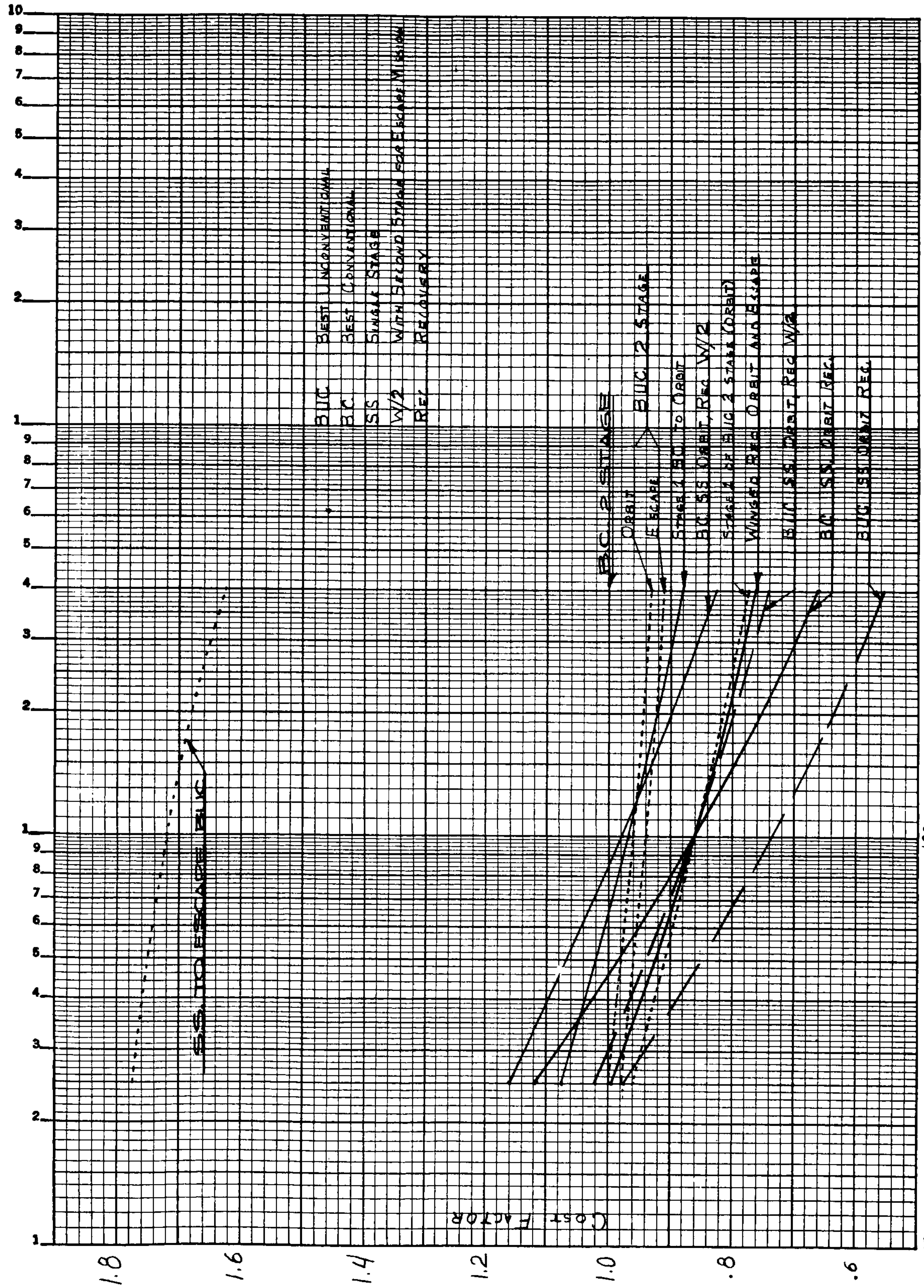
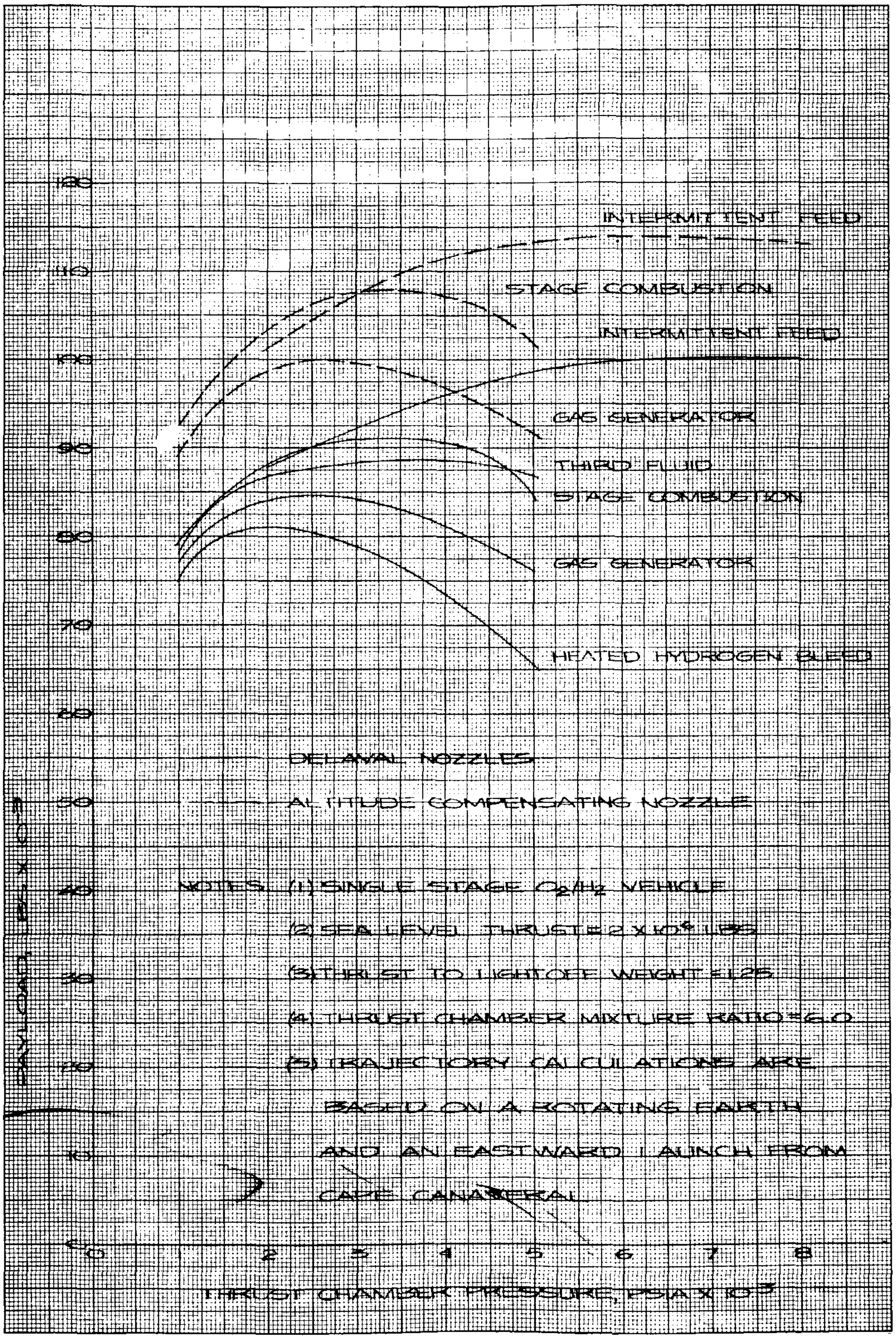


Comparison of Conventional and Unconventional Configurations

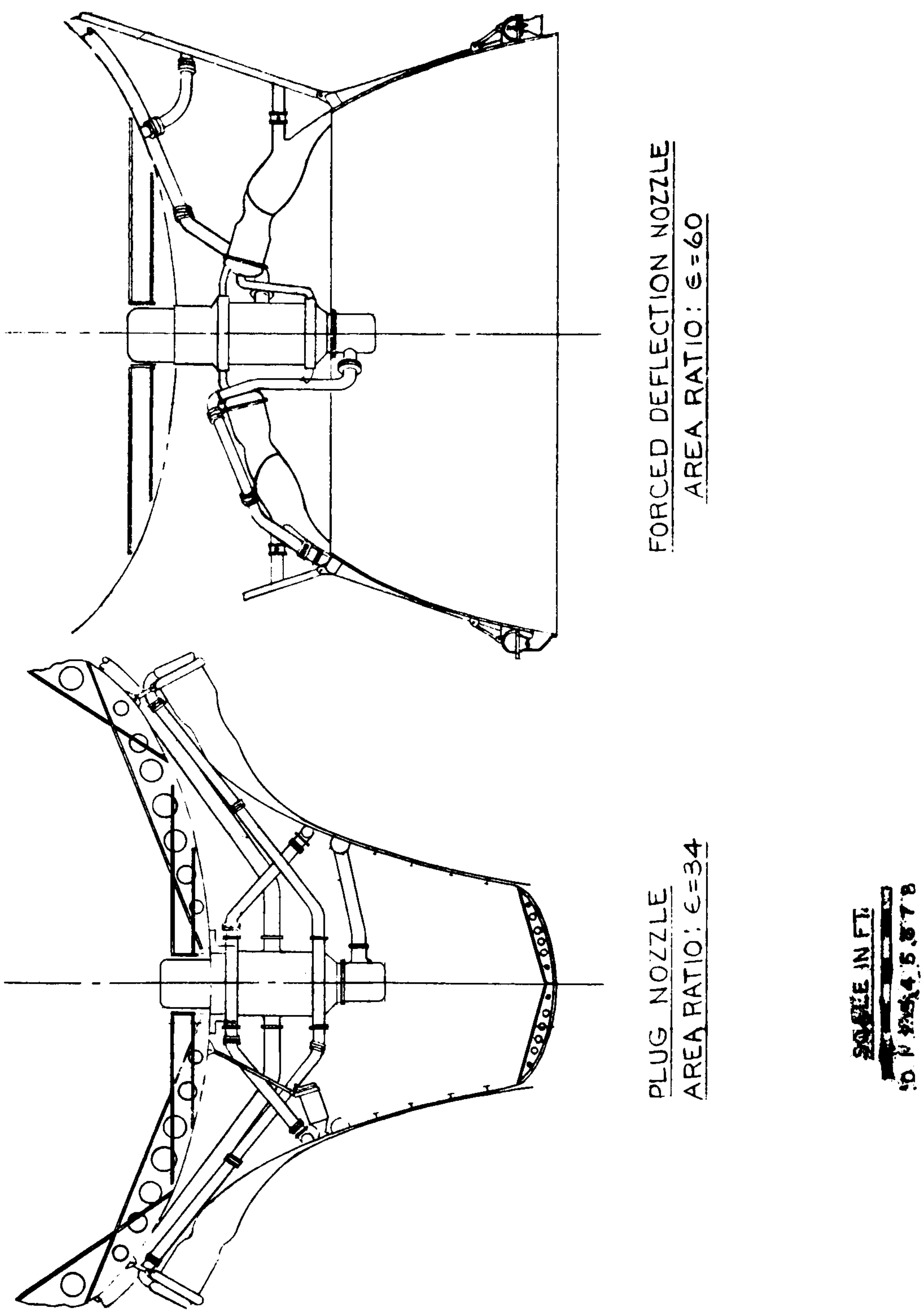
Figure III B-7

[redacted]



300-nm Payload vs Thrust Chamber Pressure for Various Engine Cycles

Figure III B-8



Two-Million-lb Thrust Engine Comparison

Figure III B-9

III, Summary (cont.)

C. TASK 2

1. General

The objective of Task 2 was to combine the features found to be beneficial during the Task 1 investigation into an engine-airframe configuration and determine the best operating parameters (i.e., chamber pressure, mixture ratio, area ratio, etc.). As a result of the Task 1 effort, further work with the airframe was not considered necessary. Therefore, Task 2 effort was concentrated upon the engine system. The following features, found to be beneficial during Task 1, were combined and the resultant engine system is shown as Figure III C-1:

a. Forced-deflection nozzle
b. Staged combustion cycle
c. Higher than conventional chamber pressure
d. Pump inducers
e. Large element injectors
f. Rigid engine-airframe integration
g. Thrust-vector control by secondary gas injection
h. Multiple discrete throat combustors, single turbopump and single nozzle.

2. Engine Descriptions

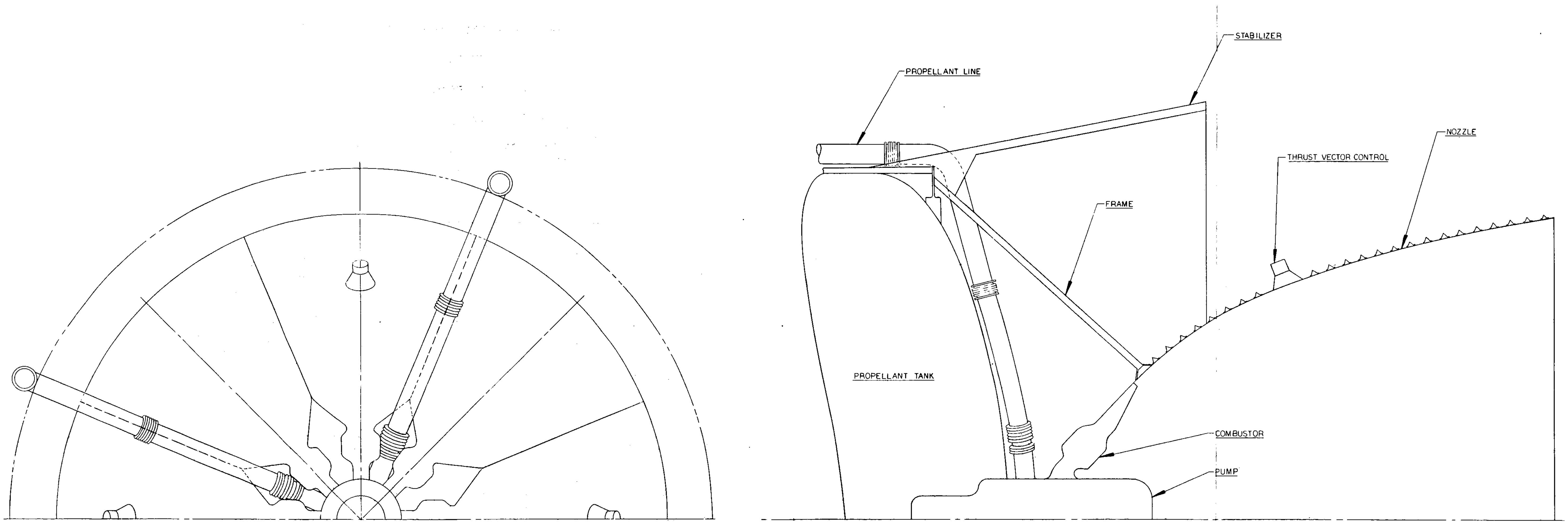
The advanced engine is shown in Figure III C-1. This engine incorporates eight combustors, circumferentially situated around the fuel turbopump and the oxidizer turbopump, which are located on the vehicle centerline. These eight combustors feed into a single forced-deflection nozzle through eight transition sections that provide the supersonic gases smooth transition between the circular throat and main expansion section.

III, C, Task 2 (cont.)

The pumping system consists of a hydrogen turbopump (top), an oxygen turbopump (bottom), and eight gas generators, circumferentially located around the turbopumps to furnish hot gas for turbine drive. This engine incorporates a staged combustion cycle; therefore, the turbines exhaust directly into the main combustors. Ambient air for altitude compensation enters the center portion of the engine between the transition nozzles. Thrust takeout to the airframe is provided though a truncated conical structure which attaches to the engine at the main nozzle near the circle where the transition nozzles and main nozzle are joined. Thrust-vector control is provided by high pressure turbine drive gases ducted, as shown, and injected radially inward through small control nozzles. Engine specifications are shown in Table III C-1. Operation of the engine is as follows.

Propellants enter the turbopumps through the suction lines shown. After being pumped to approximately 4,100 psia, 70% of the hydrogen flows into the gas generators where it is combusted with sufficient oxygen to provide gases at a mixture ratio of 1.1 for turbine drive. After passing through the turbines, this gas is ducted to the main combustor through the injector manifold. The remaining 30% of the hydrogen is passed through the thrust chamber cooling jacket, and then into the injector manifold where it is mixed with the turbine exhaust gases. The oxygen not mixed with the hydrogen in the gas generator is injected directly into the main combustor through the main injector and burned with the gas mixture injected from the injector manifold.

Engine start, steady state, shutdown, and reduced thrust control are provided by two valves in each propellant feed system. One pair of valves controls the flow of propellants to the gas generators while the other pair of valves controls the flow of propellants to the main combustors and the coolant jacket.



Advanced Engine Configuration

Figure III C-1

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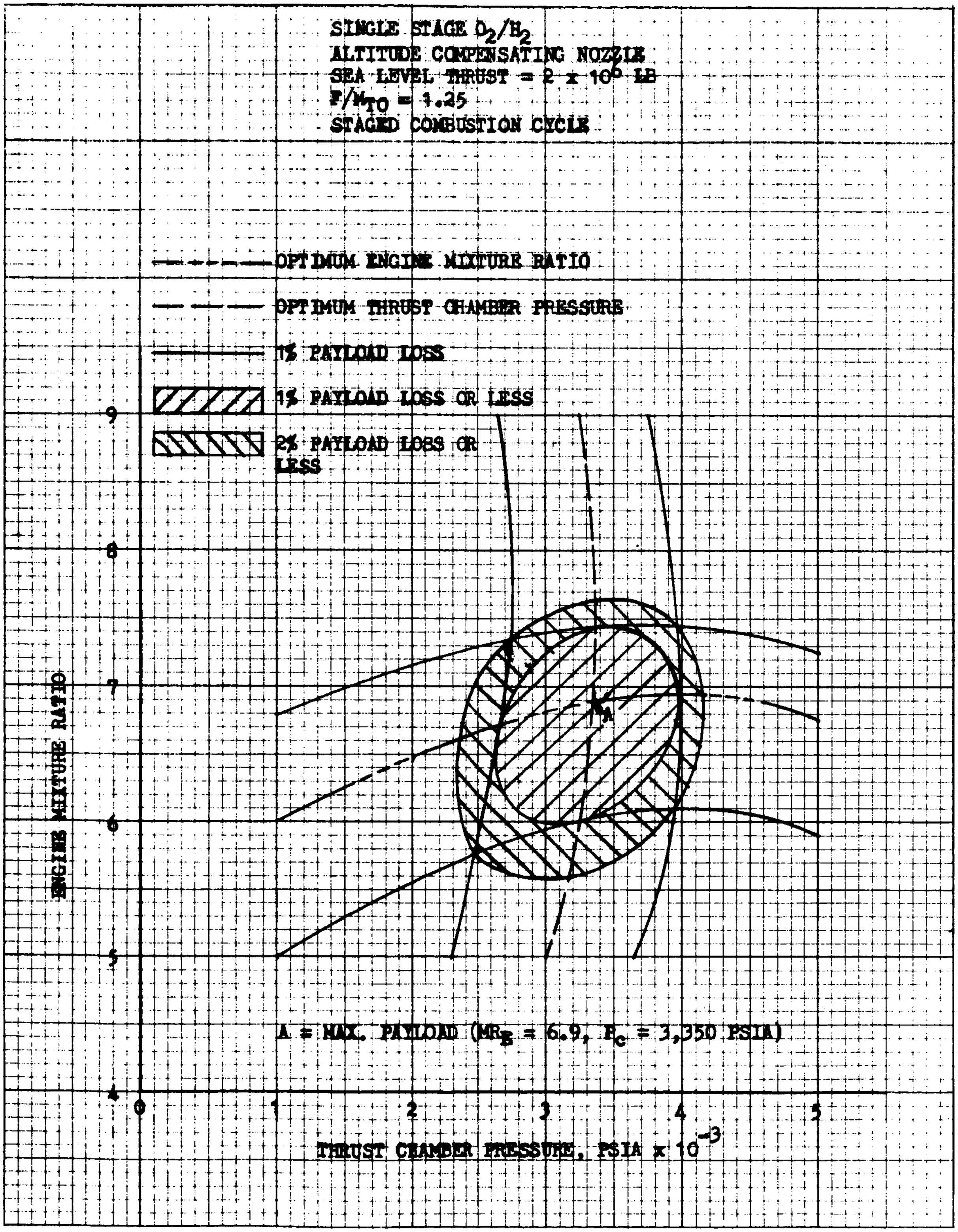

Optimum Engine Mixture Ratio and Thrust Chamber Pressure

Figure III C-2

[redacted]

III, C, Task 2 (cont.)

Thrust-vector control for the engine is furnished by four variable-flow, turbine-exhaust-supplied, secondary injection nozzles that direct flow into the lower skirt section of the main forced-deflection nozzle.

The turbopump assembly consists of a two-stage hydrogen pump and a single-stage oxygen pump driven by two parallel flow turbines and an oxygen pump inducer.

The thrust chamber combustors were designed with a contraction ratio of 3.7 and a characteristic length (L*) of 50. This high contraction ratio provides a more compact engine and the additional benefit of higher performance.

The thrust chamber regenerative-coolant circuit is composed of molybdenum tubing one-third of the way down the main nozzle and steel tubing for the remainder of the nozzle. In addition, the tubes are coated in the region near and at the throat. Some film cooling may also be required to completely solve the heat transfer problem but this has not been provided in the engine described in the previous paragraphs. It is in this heat-transfer area that much feasibility effort is required.

3. Parameter Selection

a. General

The major engine parameters are:

(1) Engine mixture ratio, 6.0
(2) Nozzle area ratio, 125
(3) Thrust chamber pressure, psia, 2,500.

III, C, Task 2 (cont.)

These parameters were selected on the basis of maximum payload to 300-nm orbit for a single-stage vehicle. A complete description of the analysis along with assumptions is contained in Section III, A, 3, of Volume 3. The analysis was divided into two phases. Chamber pressure and mixture ratio were analyzed simultaneously using a nozzle area ratio established by fixing the exit diameter equal to the vehicle diameter. Then, nozzle area ratio was analyzed at the selected values of chamber pressure and mixture ratio.

b. Chamber Pressure and Mixture Ratio

The effect of chamber pressure and mixture ratio upon single-stage vehicle payload is shown in Figure III C-2. Note that maximum payload appears to occur at a chamber pressure of 3,450 and a mixture ratio of 6.9. Design effort for the major engine components was initiated prior to completion of this analysis based upon a mixture ratio of 6.0 and a chamber pressure of 2,500 psia. As shown in Figure III C-2, very nearly optimum payload results if these parameters are used. Therefore, these parameters were not changed.

4. Nozzle Area Ratio

Using the thrust chamber pressure and mixture ratio selected in the previously indicated analysis, nozzles of various area ratios and contours were analyzed. Each nozzle contour is characterized by different exit half-angles depending upon the area ratio at which it is terminated. The results of this analysis, wherein single-stage-to-orbit payload was shown in relationship to the nozzle area ratio for various exit half angles, is presented as Figure III C-3. The highest payload results for an area ratio of 150 and an exit half-angle at 6°.

III, C, Task 2 (cont.)

However, this analysis neglected wall friction effects. Also there is increased difficulty in nozzle cooling as nozzle surface area becomes larger. Nozzle surface area increases with increasing area ratio and decreasing exit half-angle. Therefore, a somewhat lower area ratio and an increased exit half-angle than those shown to be optimum were selected. The selected nozzle area ratio of 125 and exit half-angle of 10° result in a performance loss of approximately 1% payload (Figure III C-3).

(End III C)

[redacted]

TABLE III C-1

OXYGEN/HYDROGEN TWO-MILLION-lb THRUST ADVANCED ENGINE CONCEPT ENGINE SPECIFICATIONS

Engine	Sea Level/Vacuum
Ambient pressure (psia)	14.7
Acceleration (a/g)	1.25
Specific impulse, instantaneous (sec)	386.5/450
Engine mixture ratio, $\dot{w}_o/\dot{w}_f$	6.0
Total engine thrust (lb)	$2x10^6/2.33x10^6$
Thrust Chamber	
Thrust (lb)	$2x10^6/2.33x10^6$
Thrust chamber pressure, plenum total (psia)	2,500
Thrust chamber pressure, injector face (psia)	2,580
Specific impulse, instantaneous (sec)	386.5/450*
Thrust coefficient (plenum)	1.665/1.94
Characteristic velocity (ft/sec)	7,470
Effective exhaust velocity (ft/sec)	12,440/14,070
Nozzle exit pressure (psia)	14.7/1.135
Nozzle pressure ratio, P_c/P_e	170/2,200
Total propellant flow rate, (lb/sec)	5,175
Mixture ratio, $\dot{w}_o/\dot{w}_f$	6.0
Fuel flow rate (lb/sec)	739
Oxidizer flow rate (lb/sec)	4,436
Throat area per chamber, 8 chambers (in.^{2})	60.0
Throat diameter of each chamber (in.)	8.74
Nozzle area ratio	125
Nozzle efficiency, λ	0.965/0.974
Combustion efficiency, η_c	0.98

* Specific impulse without the turbine in the system = 388/451.5

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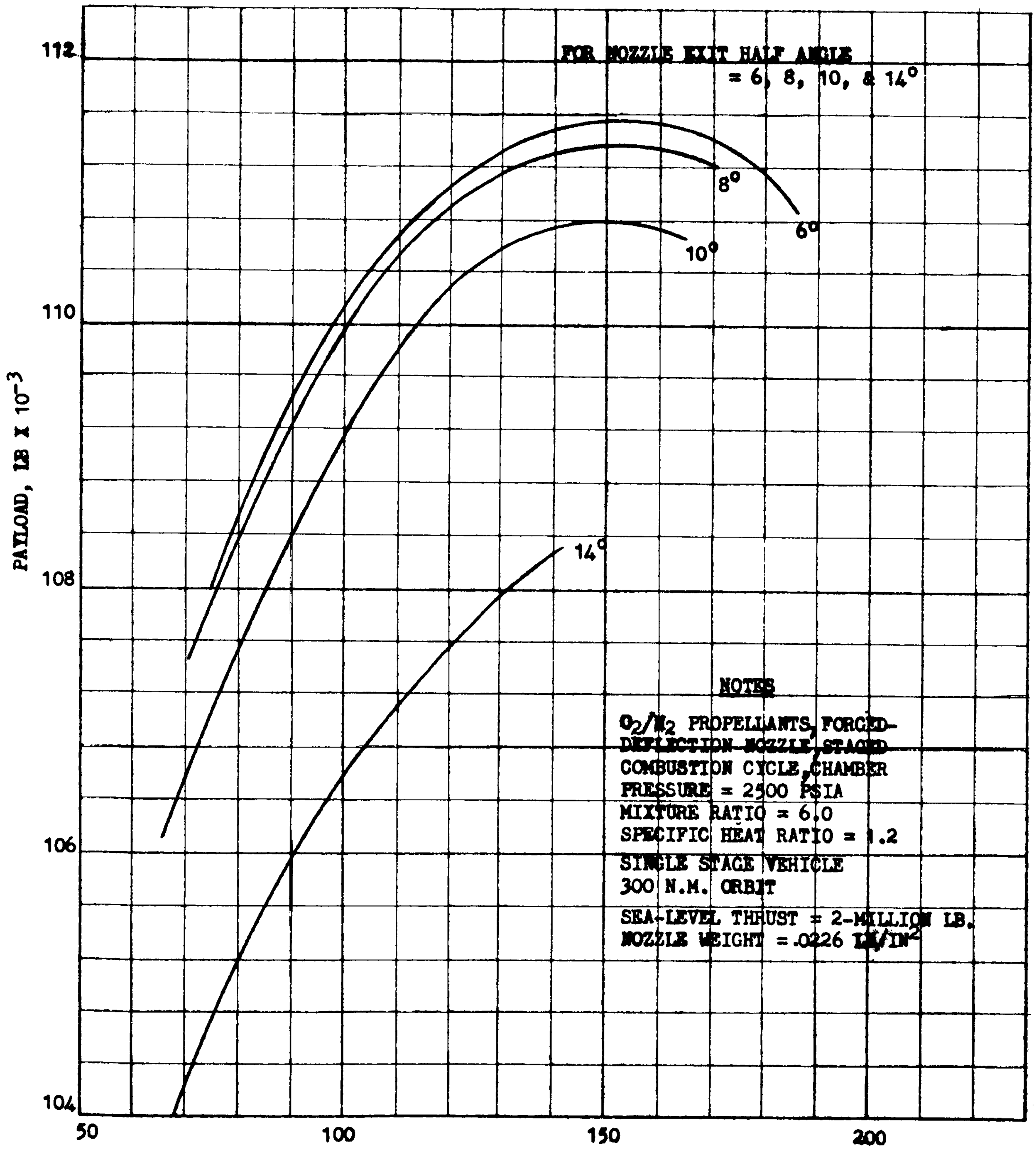

Nozzle Area Ratio Optimization

Figure III C-3

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III, Summary (cont.)

D. TASK 3

1. General

The purpose of Task 3 was to outline a detailed series of experimental and analytical programs to evaluate feasibility or provide the technology needed to make the unconventional engine a reality.

a. The final unconventional engine design generated during Task 2 was analysed to determine the areas wherein feasibility programs should be undertaken to provide a firm basis for designing new, large unconventional engines. These areas are:

(1) Performance of forced-deflection discrete throat nozzle

(2) Operating characteristics of staged combustion

(3) Large element injector performance

(4) Heat-transfer characteristics at high pressure for oxygen/hydrogen engines

(5) High-pressure hydrogen pump

(6) Staged combustion engine cycle

(7) Propellant valve concept

(8) Thrust-vector control by secondary gas injection

(9) Staged combustion with oxidizer-rich and fuel-rich gas generators

(10) Inducer pump

(11) Thrust chamber performance at high pressure

(12) Step-change in thrust for single-stage-to-orbit applications.

III, D, Task 3 (cont.)

b. Examination of current feasibility contracts show that some of the indicated feasibility areas are undergoing adequate investigation. Taking this into consideration, five new feasibility programs were formulated, as follows:

(1) Large-scale forced-deflection thrust chamber assembly segment

(2) Advanced "breadboard engine"

(3) Control components evaluation

(4) Cooling techniques for high-pressure oxygen/hydrogen engines

(5) Thrust-vector controls evaluation.

A brief summary of the recommended programs follows.

2. Large-Scale Forced-Deflection Thrust Chamber Assembly Segment

Main program objectives include: the demonstration of a large-scale oxygen/hydrogen forced-deflection thrust chamber segment operating at 2,500 psia chamber pressure using staged combustion, and obtaining data useful in the design of large, unconventional oxygen/hydrogen engines. Engine operation will be demonstrated at rated and reduced thrust. Secondary objectives will be to demonstrate thrust-vector control by secondary gas injection, and to re-evaluate demonstrated design concepts on the basis of program results.

3. "Breadboard Engine"

The purpose of the "breadboard engine" program is to demonstrate an engine using the staged-combustion engine cycle operating at a selected chamber pressure. In addition, the program provides continued engine design

III, D, Task 3 (cont.)

and analysis to integrate all test results and new concepts into the final engine design to be recommended at the end of the feasibility phase of development. This program would be best conducted as a follow-on to the large-scale forced-deflection thrust chamber assembly segment program because much of the hardware used in this program would be applicable to the "breadboard engine" program.

4. Controls Components Evaluation

The controls component program includes a design and analysis evaluation to describe control requirements and specifications for advanced engine systems. It also encompasses a feasibility program to demonstrate the operation of the main propellant controls for the advanced engine design.

5. Cooling Techniques for High-Pressure Oxygen/Hydrogen Engine

This program has as its objective the determination of the best cooling method for the advanced engine. It will be accomplished as follows.

Analytical and design investigations will be conducted to determine applicability of various cooling techniques to the advanced engine thrust chambers. Heat-transfer data will be established, as required, in laboratory tests. Hardware incorporating the cooling method shown to be best by the design and analysis phase, will be designed, fabricated, and tested. This will demonstrate the adequacy of the cooling method shown to be best in the design and analysis phase.

III, D, Task 3 (cont.)

6. Thrust-Vector Control Evaluation

Major objective of this program is the evaluation of thrust-vector control by secondary fluid injection for the advanced engine. The program consists of design, analysis, and testing.

In the design and analysis phase, available test data and theory for secondary fluid injection will be reviewed to determine the best available design criteria. A scale model will be designed to obtain functional dependency of variables not investigated in previous testing. This model will be designed to duplicate conditions in the advanced engine in so far as practicable. The model will be fabricated and tested to determine dependency of thrust-vector control effectiveness upon variables for which insufficient data exists. Results from the test phase will be used to design an optimum thrust-vector control system for the advanced engine.

III, Summary (cont.)

E. TASK 4

The objective of Task 4 was to establish factors for determining cost in relationship to thrust levels for systems up to 24-million-lb thrust. A great deal of this effort was necessarily accomplished during Task 1 with the establishment of the cost model for evaluating various concepts. Therefore, the major portion of Task 4 was devoted to determining the effects of thrust level upon cost for the advanced-type of engine.

To accomplish this objective, the two million-lb thrust advanced engine was scaled directly to 24-million-lb thrust and designs prepared. The only exception was the selection of 16 thrust chambers and gas generators instead of eight so that thrust chamber testing could be accomplished using existing facilities. Development and production costs, along with facilities costs, were then estimated for both the two-million- and 24-million-lb thrust engines. Scaling factors were evolved from this data.

Costs are most conveniently scaled by means of log-log plots of the cost data. Therefore, each scaling factor is presented as the exponent of the general equation:

$$C = A(M)^b$$

where C = cost, dollars

A = constant

M = independent variable

b = cost scaling factor

III, Summary (cont.)

The estimated cost scaling factor, b, along with the independent variable, M, for each item in the cost model are presented in Table III E-1.

Results of the Task 4 effort showed that large turbopumps can be developed and are superior to clustered turbopumps with respect to both cost and reliability. Thus, the basic philosophy of using single turbopumps evolved during Task 1 was reaffirmed at the 24-million-lb thrust level. Also, the use of clustered complete engines is unnecessary at the 24-million-lb thrust level.

TABLE III E-1

COST SCALING FACTORS

	Independent Variable	Scaling Factor, b
Engine development (Engineering, Test, Fabrication, and R and D Tooling)	Thrust	0.37
Propellants required for engine development	Thrust	0.52
Facility cost	Thrust	0.64
Engine GSE	Thrust	0.20
Engine production tooling	Thrust	0.20
Airframe development	Thrust	0.22
Engine production	Thrust	0.74
Engine acceptance testing	Thrust	0.34
Airframe production	Tankage Volume	0.45
Stage transportation	Tankage Volume	0.17
Stage launch operation	Thrust	0.14
Range time	(Cost is Independent of Size)	-
Propellants for engine calibration, stage acceptance tests, and flight	Thrust	1.0

III, Summary, (cont.)

F. TASK 5

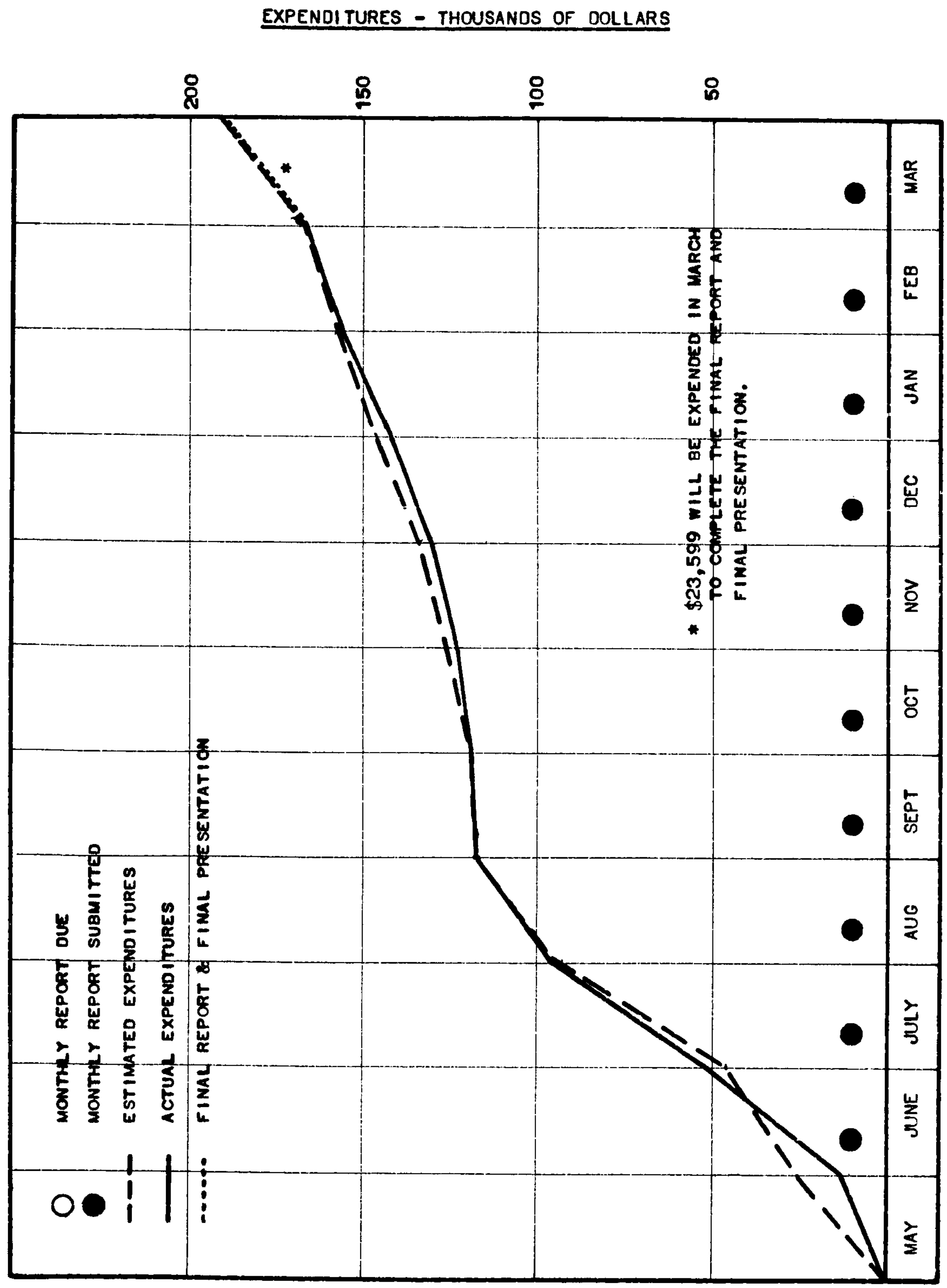
The objective of Task 5 was to investigate the feasibility of the advanced engine for second-stage applications. This objective was kept in mind throughout the Task 2 design study so that the resultant engine is entirely suitable, without modification, for upper-stage application.

IV. FISCAL REPORT

Figure IV-1 is the complete program summary, showing estimated expenditures in relationship to actual expenditures.

Total budgeted manhours for Task 1	10,310
Total budgeted manhours for Tasks 2 to 6	5,459
Total manhours expended	15,769
Total budgeted dollars for Task 1	95,926
Total budgeted dollars for airframe subcontract (The Boeing Company)	18,271
Total budgeted dollars for Tasks 2 to 6*	53,204
Total budgeted dollars	167,401
Total dollars committed for final report and final presentation	23,599
Total dollars expended	191,000

*Excluding Task 6 current commitments.



Design of a Large Unconventional Liquid Propellant Rocket Engine and Vehicle, Performance Chart

Figure IV-1